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EVALUATION OF THE COMPLIANCE OF AIRCRAFT WING FLAP WITH USE OF THE FRACTURE MECHANICS

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Abstract

Research in the field of fracture mechanics and determination of material characteristics are used for practical purposes, such as the assessment of static and dynamic strength of structural components, analysis of their fatigue life or extending the life span of their operation. A structural component, considered to be safe from fatigue cracking point of view, was investigated and results were presented in this article. In particular, an analysis was made to determine the stress intensity factor for the cracked wing flap construction, based on static and fatigue tests, using the Irwin-Kies theory. The flap with a service crack was subjected to fatigue tests with a load similar to the one registered during flight measurements. The flap without a service crack was subjected to static tests, after cutting the cracks of specified lengths and shapes (similar to the service crack) in the skin of the flap. The article presents changing the length of the flap crack in subsequent load cycles, change in the maximum values of force and the crack opening displacement in subsequent load cycles, dependence of P-COD in the first and second stage of fatigue testing of the wing flap, dependence of the wing flap compliance on the length of the crack and experimentally determined dependence for wing flap. The occurrence of a flap crack up to approximately 230 mm does not cause a significant growth of the stress intensity factor.

Keywords: *airplane wing flap, crack growth, stress intensity factor, Irwin-Kies theory*

1. Introduction

A well-designed structural component should not be distressed during the operation before the specified / approved time. The degree of its wear ought to be subjected to effective control so that the component can be safely operated and possible to replace before the wear reaches a critical level and structural damage occurs [2-3, 10-12]. Analysis of structural components distressed during operation is good and sometimes the only opportunity to confront theoretical and laboratory research with practice. Usually the stress intensity factor and the relationships of crack opening displacement *COD* vs. crack length a is known for the single notch specimens [1, 6, 7]. However, some guide-boxes or standards do not treat this type of specimens as standard specimens for fatigue tests, so the relationships of crack length vs. compliance do not have commonly. The knowledge of this function is primary need for use of computer software in fatigue testing systems. This allow to be crack length automatically measured based on crack opening displacement *COD* vs. loading force P data logging. Software of the majority of testing machines uses the dependence of the non-dimensional crack length *a/W* on the non-dimensional compliance *U* defined with the following formula [6-10]:

$$U = \frac{1}{1+\sqrt{BE\frac{COD}{P}}}, \tag{1}$$

where *B* is specimen's width, *E* is Young modulus of the specimen's material.

This article presents the results of research on the wing flaps of the Mig-21 aircraft carried out on the laboratory stand shown in Fig. 1 [4, 5, 12], which used a flap with a service crack in the bracket area and a flap without a service crack in which the crack covering was cut with specific lengths and shape similar to the service crack. The flap with a service crack was subjected to fatigue testing with a load similar to that measured during flight measurements. The flap without a service crack was subjected to static tests, with crack mechanically cutting in the flap cover with the specified lengths and shape similar to the service crack.

2. Methodology

The test flap is mounted in an inverted position on the brackets ensuring free rotation around the axis of the own damper. The flap hoist was mounted on a vertical angle, so that the hoist's cylinder axis was in relation to the plane of the bottom cover of the flap at an angle analogous to that in the position of maximum operational loads. The hoist was hydraulically powered from the stand using for testing hydraulic aggregates of airframe aircraft.

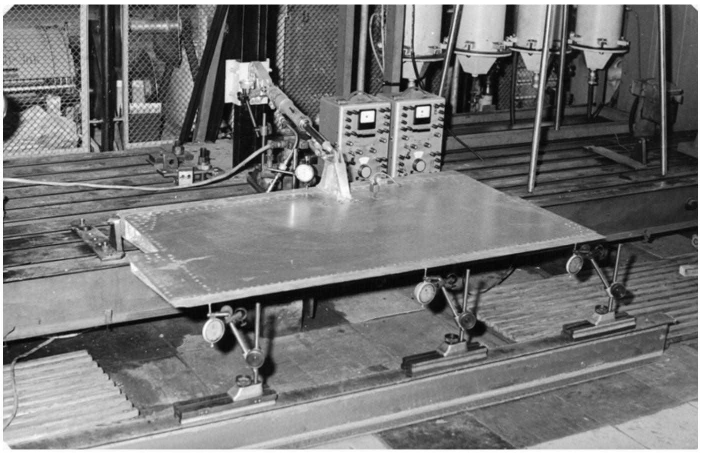

Fig. 1. View of a wing flap mounted on the test bench [12]

The force on the hoist was measured using a strain gauge (T1) bonded on its handle closer to the hoist − Fig. 2. The load of the hoist was realized by means of sandbags imitating its operational aerodynamic load − the weight of the bags was chosen so that the value of the force on the hoist was equal to the value measured during the flight. The crack length in the flap sheathing was measured initially (for small cracks) using strain gauges (T2-T7) glued on the crack propagating path, and for larger lengths of cracks by means of observation with the use of a magnifying glass − for the crack length the sum of the cracks distance from the flap hoist plane of symmetry was assumed. The crack opening displacement was measured using a CG gauge.

The analysis aimed at determining the value of the stress intensity factor for the damper construction was carried out, using the Irwin-Kies theory described below, which will also allow determination of the *da/dN-ΔK* dependence. When a cracked plate of thickness *B* and width

W suffers an infinitesimally small crack growth, given the force P, the deformation energy G released per unit thickness is expressed by the Irwin-Kies formula [6]:

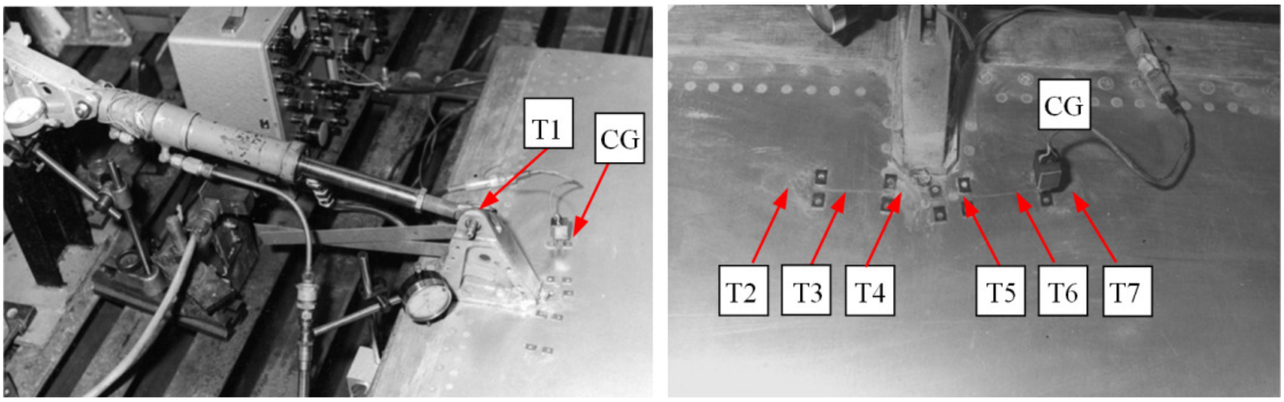

Fig. 2. View of the hoist of flap, locations of the strain gauges and gap crack detector [5,12]

$$G = \frac{P^2}{2B}\frac{\partial C}{\partial a}, \tag{2}$$

where C is the compliance COD/P of the plate, the ratio of the crack opening displacement to the loading force.

Between the stress intensity factor K (assuming that it can be described by an analogous relation as for compact samples, subjected simultaneously to stretching and bending) and energy G there is a dependence:

$$K = \frac{P}{B\sqrt{W}}Y\left(a/W\right) = \begin{cases} \sqrt{GE} & \text{for } \textit{plane stress}, \\ \sqrt{\dfrac{GE}{1-\upsilon^2}} & \text{for } \textit{plane strain}, \end{cases} \tag{3}$$

respectively for a plane state of stress and a plane state of deformation.

There are also:

$$Y\left(a/W\right) = \frac{KB\sqrt{W}}{P} = \begin{cases} \sqrt{\dfrac{1}{2}\left(\dfrac{\partial(CEB)}{\partial(a/W)}\right)} & \text{for } \textit{plane stress}, \\ \sqrt{\dfrac{1}{2(1-\upsilon^2)}\left(\dfrac{\partial(CEB)}{\partial(a/W)}\right)} & \text{for } \textit{plane strain}. \end{cases} \tag{4}$$

If determining the derivative of the compliance (modified according to the above formula), it is possible to determine the shape function for the stress intensity factor $Y(a/W)$.

3. Results

The measured lengths of the crack propagating in the flap are shown in Fig. 3, where can be distinguished two stages obtained: with strain gauges (i.e. until the crack tip reached the individual strain gauges − stage I) and as a result of visual observation (i.e. when the crack length exceeded the range of attached strain gauges, approximately 130 mm − stage II). In the second stage of research, the level of loads applied to the flap was increased in order to shorten the duration of tests; hence, the rate of crack growth is also higher.

In both test phases, the corresponding sections of the obtained a-N curve are approximately linear, which indicates a stable crack growth throughout the test.

The magnitude of the maximum forces on the flap hoist and the corresponding crack displacement measured during fatigue tests are presented in Fig. 4. It can be seen that in the initial test period (up to about 4,000 load cycles), the value of force dropped significantly and the opening displacement of the crack slightly decreased. This was the effect of an intense degradation of the adhesive joint of the honeycomb filler with flap sheathing. The mentioned phenomena was the cause of the gradual change of the hoist's axle configuration relative to the flap surface, significant weakening of the internal structure of the flap. Consequently local relieve the flap sheathing and relatively little damage (the crack in the sheathing only increased about 2.5 mm) took place.

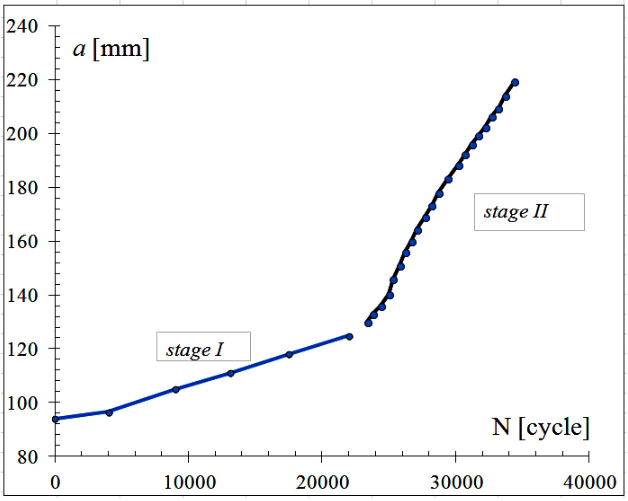

Fig. 3. Changing the length of the flap crack in subsequent load cycles

In the next 18,000 load cycles, the crack increased faster (by about 30 mm), and crack opening displacement was only growing − which indicates intensive destruction of the flap sheathing, possibly with a small influence of the honeycomb filler on the local deformation of the sheathing. The increased level of load in stage II of the test significantly accelerated the propagation of the crack and the crack opening displacement increased more rapidly. The crack reached a length of about 220 mm (*2a* equal about 400 mm), with a crack opening displacement of 0.7 mm − which corresponds to the situation that should not occur in such a scale in operating conditions. The measurements were completed at this stage. However, the flap was subjected to further loads with the intention of obtaining a state of it destruction. After 110,000 cycles, a crack size of 90% of flap width was achieved, reaching the extreme rivets on the edge ribs (Fig. 5). The crack opening displacement rise to about 2.8 mm. Nevertheless, the flap continued to carry the full load − which indicates a considerable resistance of the airplane flap structure to a fatigue crack in its sheathing.

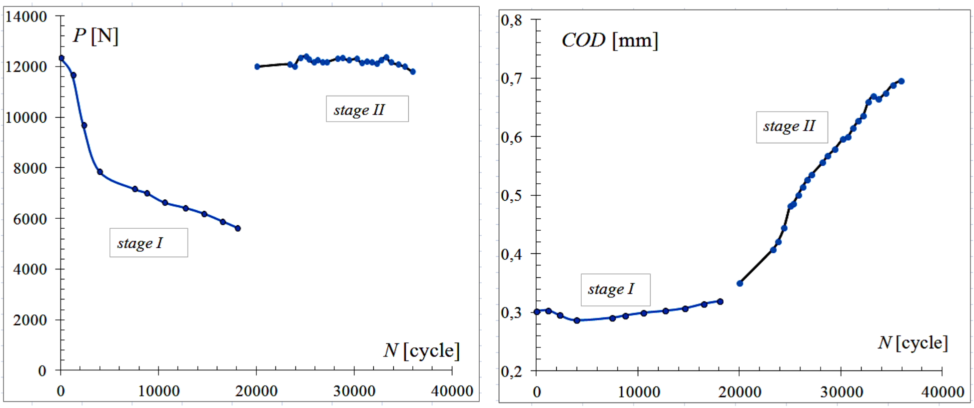

Fig. 4. Change in the maximum values of force and the crack opening displacement in subsequent load cycles

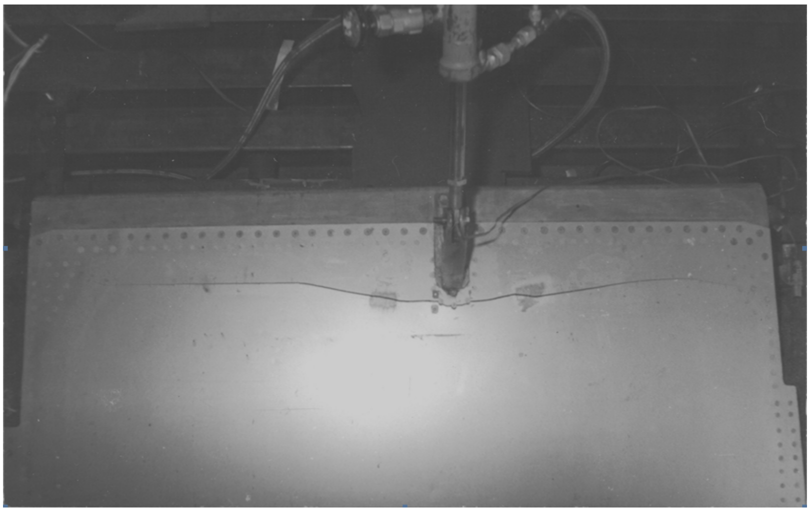

Fig. 5. View of the flap crack with a size of approx. 90% of its width [12]

A statement of the *P-COD* parameters measured in the first and second stage of fatigue tests and the method of determining the compliance *C* for each of the curves are shown in Fig. 6. The compliance, plotted against the crack length corresponding to the individual *P-COD* curves (Fig. 7), was used to calculate the stress intensity factor – in accordance with formulas (3) and (4).

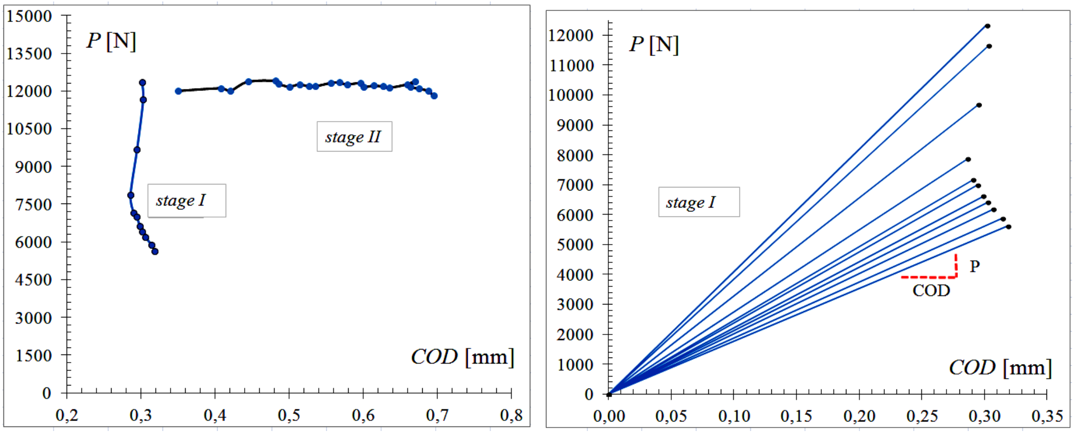

Fig. 6. Dependence of P-COD in the first and second stage of fatigue testing of the wing flap

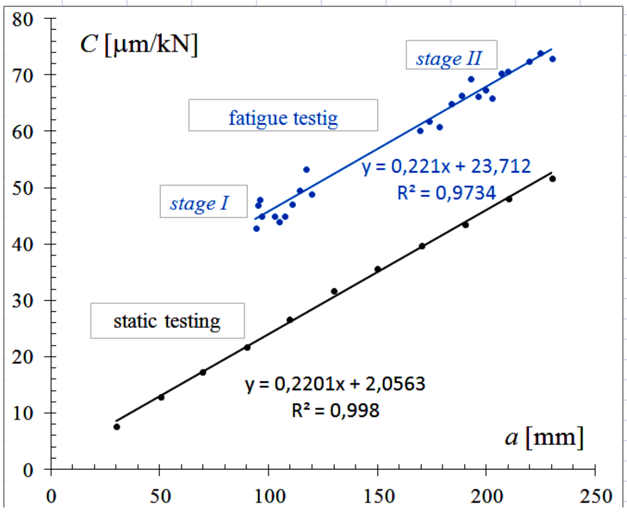

Fig. 7. Dependence of the wing flap compliance on the length of the crack

The abovementioned results were compared to analogous measurements on the flap, which was not subjected to fatigue load. The second flap was used, in which mechanical cracks of fixed sizes were cut (shape and sizes close to the crack in the first tested flap) and loaded statically on the next set levels of load. The *COD-P* curves determined for individual crack lengths allowed to determine the static compliance of a wing flap. Fig. 7 shows both determined relationships of compliance C (i.e. fatigue and static) as a function of the crack length a. The compliance of a fatigue-tested flap is about two times greater compared to obtain for a statically tested flap. The probably reason could be the different degree of destruction of the internal structure of the honeycomb filler and its adhesive joints during fatigue and static tests, as well as the difference in crack geometry and its stiffness in these studies.

Despite these differences, both relationships $C=f(a)$ are characterized by the same slope of linear regression curves − and thus also the same derivative $\partial C/\partial a$, which is used in formulas (4). Compliance C for individual crack sizes (formula (4)) and its differentiation allow to determine the shape function of the stress intensity factor $Y=f(a/W)$ and the stress intensity factor applicable to the geometry of the wing flap − Fig. 8. In the tested range of variations of dimensionless crack length a/W (up to the value of 0.35) both these functions remain stable with small value fluctuations. The occurrence of a crack with dimensions up to 230 mm does not cause a significant increase in the stress intensity factor − Fig. 8b.

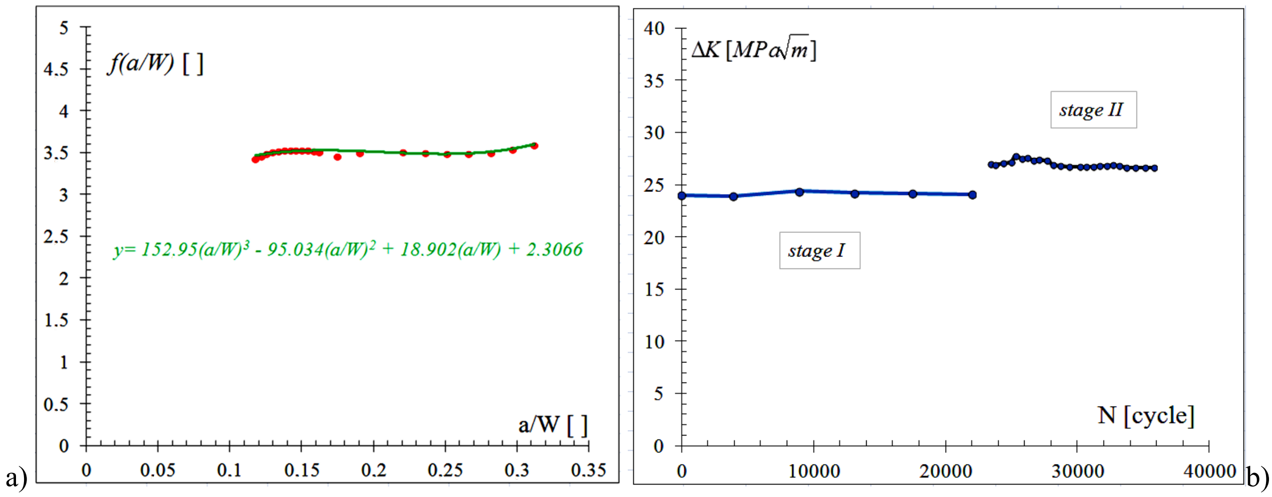

Fig. 8. Experimentally determined dependence for wing flap:
(a) of the dimensionless shape function of the stress intensity factor and
(b) of the stress intensity factor

The propagation curve $da/dN=f(\Delta K)$, *having regard to* the variability of the propagation rate da/dN of fatigue crack in the sheathing of a wing flap (calculated, separately for each test stage, as a derivative of the course of the curve *a-N* from Fig. 3) and the variability of its corresponding value ΔK range of the stress intensity factor, has the form presented in Fig. 9a. The black points are results of the calculation based on regression of two polynomials of both *a-N* curves, and red points remain the results based on partial 7-points linear regressions [8]. The comparison of this curve with the crack propagation curves obtained for testing samples with a central notch M(T) made of Al 2024 sheets with a thickness of 1.2 mm is shown in Fig. 9b. Due to testing M(T) samples for the value of the stress ratio $R = 0.2$; 0.5 and 0.8 and the wing flap for $R = 0$ (the minimum force on the flap hoist during the tests was 0 N) − a good match between both test results is apparent.

The magnitudes of loads, the duration of the flap test and the rate of crack growth in the flap sheathing indicate an advanced, critical crack propagation process − which is also confirmed by the data presented in the Fig. 9.

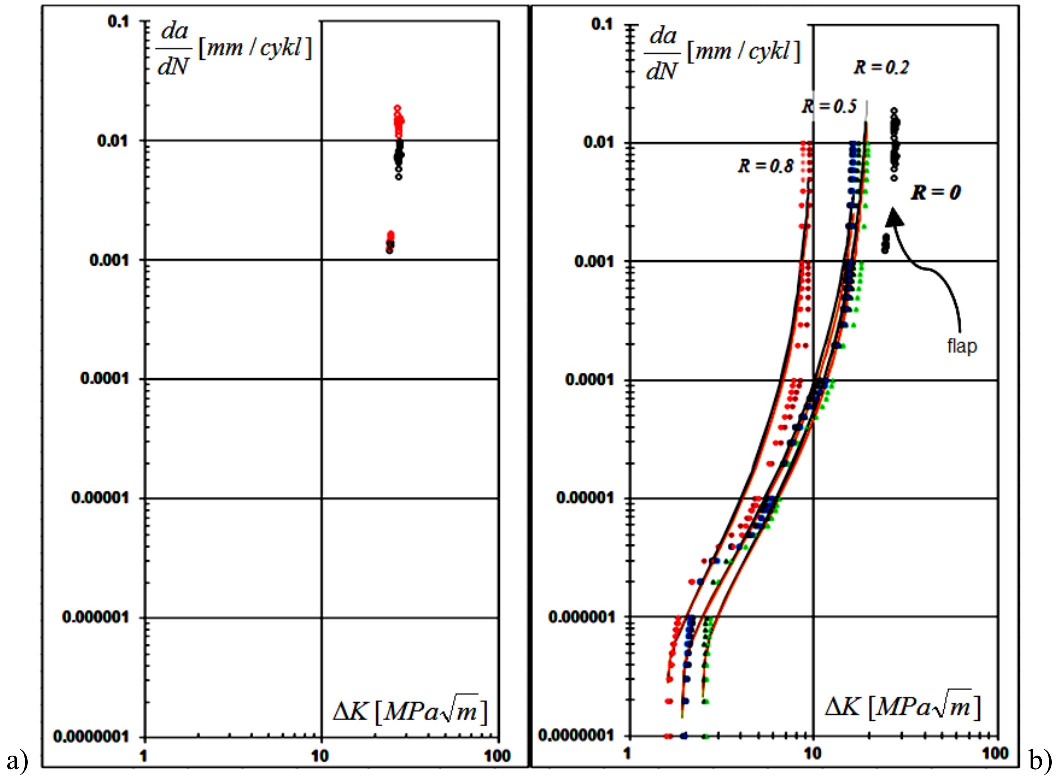

Fig. 9. Experimentally determined for both test phases, the dependence da/dN=f(ΔK) for the wing flap (a) and the comparison with the propagation curves obtained for testing of the M(T) samples from aluminum sheets with a thickness of 1.2 mm [8] (b)

4. Conclusions

The fatigue crack propagation in the wing flap of the Mig-21 aircraft registered during mechanical tests proceeds without critical stage, in other words without rapid crack propagation, which leads to distortion of the flap. The investigated part sets the example of the structural component designed as fail-safe, so the occurrence of fatigue crack during its operation does not endanger to the flight safety but it can and should be monitor during current and periodic inspections of the aircraft structure.

The shape function of the stress intensity factor as well as the stress intensity factor itself for the aircraft wing flap, remain stable with small value fluctuations. The occurrence of a flap crack up to approximately 230 mm does not cause a significant growth of the stress intensity factor.

References

[1] Bukowski, L., Kłysz, S., *Compliance curve for single-edge notch specimen.* Zagadnienia Eksploatacji Maszyn, Zeszyt 2 (134), pp. 95-104, 2003.

[2] Bukowski, L., Kłysz, S., *Some problems of combat aircraft case investigation.* Physico-chemical Mechanics of Materials, Lvov, Vol. 31, No. 1, pp. 59-62, 1995.

[3] Bukowski, L., Kłysz, S., *Zmęczenie konstrukcji lotniczych.* Problemy badań i eksploatacji techniki lotniczej. Praca zbiorowa pod red. J. Lewitowicza, J. Borgonia, W. Ząbkowicza, Vol. 2, Wyd. ITWL, pp. 409-450, 1993.

[4] Bukowski, L., Kłysz, S., *Analiza rozwoju pęknięć zmęczeniowych w klapach zaskrzydłowych samolotu Mig-21*. Informator ITWL, Nr 70, pp.147-160, 1989.

[5] Bukowski, L., Kłysz, S., *Badanie współczynnika intensywności naprężeń dla klapy zaskrzydłowej samolotu Mig-21*. Informator ITWL, Nr 70, pp. 135-146, 1989.

[6] Cartwright, D. J., Rook, D. F., *Evaluation of stress intensity factor.* J. of Strain Analysis, Vol.10, No 4, 1975.

[7] Goss, Cz., Leśniewski, J., Kłysz, S., *The propagating fatigue crack length estimation based on the crack opening at the specimen edge measurement results.* 8 Int. Conf. of Fract., Kiev, pp. 60-65, 1993. J. of Theor. and Appl. Mechanics, No. 4, Vol. 30, pp. 833-842, 1992.

[8] Kłysz, S., *Badania propagacji pęknięć zmęczeniowych na przykładzie stopu Al 2024.* Wydawnictwo Naukowe Instytutu Technologii Eksploatacji – PIB, ISBN 978-83-7789-462-0, p.212, 2017.

[9] Kłysz, S., *Szacowanie trwałości wybranych materiałów i elementów konstrukcji lotniczych w zakresie rozwoju pęknięć zmęczeniowych.* Prace Naukowe ITWL, Nr 5, pp. 183, 1999, ISSN 1234-3544.

[10] Kłysz, S., *Rozwój pęknięć zmęczeniowych w materiałach lotniczych i stali konstrukcyjnej z uwzględnieniem przeciążeń*. Prace Naukowe ITWL, Nr 12, pp. 151, 2001, ISSN 1234-3544.

[11] Lipiński T., Szabracki P., *Mechanical Properties of AlSi9Mg Alloy with a Sodium Modifier*. Solid State Phenomena, Vol. 223, pp. 78-86, 2015.

[12] Sprawozdanie ITWL „*Badania wytrzymałościowe klap samolotu Mig 21*”, Nr BT 8311/I, Warszawa, 1989.

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